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14. ABSTRACT There has been a growing move in the aerospace industry and a growing need in the Department of Defense to make space more responsive and cost effective. Instead of taking years to design and deploy a new satellite, the goal is weeks or even days. To meet this challenge, the methodologies used to design, manufacture, test, launch, and deploy satellites must radically change. One of the most challenging aspects of this problem is the satellite's Thermal Control System (TCS). Traditionally, the TCS is vigorously designed, analyzed, and optimized for every satellite mission. The ideal TCS for responsive space would be robust and modular with an inherent plug-and-play capability. Unfortunately missions, payloads, and requirements for ORS are still somewhat nebulous. To provide a baseline for TCS design and to help bound the problem for the development of thermal plug-and-play systems, the range of external and internal heat loads for small satellites were evaluated, and the worst hot and cold cases were identified. Using these cases, two different design schemes were investigated and compared. The first was a semi-passive conduction based scheme, and the second was a thermal switch based system. Each system was able to meet the needs of the satellite for the hot case. However, excessive survival heater power was required for the semi-passive conduction based scheme. As for the thermal switch design scheme, the need for survival heaters was virtually eliminated at the cost of added system mass and complexity. From the designs, various design parameters were evaluated, and the feasibility of a one-size-fits-most approach was assessed.

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Comparing Traditional Design Approaches to Thermal Management Design with Application to Responsive Space

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There has been a growing move in the aerospace industry and a growing need in the Department of Defense to make space more responsive and cost effective. Instead of taking years to design and deploy a new satellite, the goal is weeks or even days. To meet this challenge, the methodologies used to design, manufacture, test, launch, and deploy satellites must radically change. One of the most challenging aspects of this problem is the satellite's Thermal Control System (TCS). Traditionally, the TCS is vigorously designed, analyzed, and optimized for every satellite mission. The ideal TCS for responsive space would be robust and modular with an inherent plug-and-play capability. Unfortunately missions, payloads, and requirements for ORS are still somewhat nebulous. To provide a baseline for TCS design and to help bound the problem for the development of thermal plug-and-play systems, the range of external and internal heat loads for small satellites were evaluated, and the worst hot and cold cases were identified. Using these cases, two different design schemes were investigated and compared. The first was a semi-passive conduction based scheme, and the second was a thermal switch based system. Each system was able to meet the needs of the satellite for the hot case. However, excessive survival heater power was required for the semi-passive conduction based scheme. As for the thermal switch design scheme, the need for survival heaters was virtually eliminated at the cost of added system mass and complexity. From the designs, various design parameters were evaluated, and the feasibility of a one-size-fits-most approach was assessed.

Nomenclature

a	=	Earth albedo coefficient
$A_{\perp}$	=	area perpendicular to the sun [m]
A_{rad}	=	radiator surface area [m]
A_s	=	surface area [m]
α	=	surface solar absorptivity
ε	=	emissivity of the spacecraft
$F_{s,e}$	=	view factor between the spacecraft and the Earth
$F_{s,se}$	=	view factor between the spacecraft and the sunlit Earth
I_{EIR}	=	intensity of the Earth IR
I_{sun}	=	solar heat flux [W/m^2]
$Q_{Internal}$	=	internal heat generation [W]
σ	=	Stefan-Boltzmann constant [W/m^2-K^4]
T_s	=	average temperature of the spacecraft [K]

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I. Introduction

There has been a growing move in the aerospace industry and a growing need in the Department of Defense (DOD) to make space more responsive and cost effective. Instead of taking years to design and deploy a new satellite, the goal is weeks or even days. The drive is to extend the advantages space affords from the strategic planner to the battlefield commanders by providing the ultimate high ground. To meet this challenge, the methodologies used to design, manufacture, test, launch, and deploy satellites must radically change. For space to become operationally responsive, satellites must be easily manufactured, assembled, tested, and prepared for launch in a military depot style environment. Designs will have to be simple and robust so that Airmen play a central role rather than Ph.D.-level scientists.

One of the most challenging aspects of this problem is the satellite's Thermal Control System (TCS). Traditionally, the TCS is vigorously designed, analyzed, and optimized for every satellite mission. This "reinvention of the wheel" is costly and time intensive. The first step in the thermal design process is to determine the component temperature limits, the internal power dissipation, and the worst case environmental heat inputs. Using these inputs, the basic energy balance of the satellite is determined. Next, a simplified thermal model is developed and different TCS concepts are evaluated. At this point, the process becomes iterative. For each successive analysis, the fidelity of the thermal model is increased, and the TCS design is refined. As the design of the satellite and the components change, the thermal model must be updated. Each change must be analyzed for its effect on the TCS and the overall design of the spacecraft. Once the detailed design and thermal model are completed, they must be validated in a thermal balance test. Using the results from the thermal balance test, the design is finalized. The final proof test for the system is the thermal vacuum test. The result is a long and arduous process that will not meet the goals of ORS.

One approach to achieve the goals of ORS in the near term is to separate the design and engineering of the payload from the bus. The bus would provide a specific range of baseline capabilities to meet the needs of most missions and payloads. The goal is an 80% design solution. Any additional capability required by the payload would have to be provided by the payload itself. Integration of the bus and payload would occur through standard mechanical, electrical, thermal, and software interfaces. It should be noted that, according to this philosophy, there will be some payloads that can not be economically accommodated by the bus, and a unique system will have to be designed.

Currently, there are many companies that have developed or are developing low-cost, standard, satellite bus architectures that are specifically directed toward ORS. The disadvantage with most standardized bus development programs is that the bus eventually becomes obsolete and must be completely redesigned as new technologies are developed. One of the goals of the ORS program is the development of technologies that provide robust and flexible bus designs. For example, a space plug-and-play concept similar to PC based plug-and-play USB connectivity is being investigated for the avionics system[1]. Plug-and-play addresses the software and electrical interfaces, but other efforts are needed to address the mechanical and thermal interfaces

II. Satellite Bus Summary

Regardless of the design philosophy, a certain level of fidelity of the bus design is needed before the basic requirements for the thermal control system can be identified. Because of launch vehicle limitations, ORS missions will likely be relegated to 450 kg class satellites. Using this basic assumption, the capabilities that a small satellite bus can provide can be determined. In a previous effort, two satellite busses were sized to meet responsive space needs. Subsystem components were sized to provide a baseline capability and an upgraded one. From these components, the mass, volume, and power for each subsystem were estimated. The results were an upper and lower bounds for the design of the TCS and are only summarized here. A more detailed analysis can be found in Reference 2.

Table 1: Satellite subsystem summary for the low capability bus

Subsystem	Capability	Mass	Power	Size
		[kg]	[W]	[cm]
Attitude Determination & Control (ADC)	1°-5° attitude control	10.3	18.5	30 x 24 x 12
Telemetry, Tracking, & Command (TTC)	1 Mbs, S-band transmitter	2.8	7.4	9.8 x 9.6 x 7.2
Navigation & Guidance (NG)	12 channel GPS receiver	0.02	0.8	7.0 x 4.5 x 1.0
Command & Data Handling (CDH)	Plug-n-play USB architecture	15.2	50	34 x 25 x 20
Power Management (PM)	500 W, 3J array, PPT system	18.3	70.3	25 x 23 x 21
Structure	Al Honeycomb Panels	21.5	n/a	27 x 40.5 x 71
Propulsion	No propulsion system	0	0	0 x 0 x 0
		68.1	**147.0**	**27 x 40.5 x 71**

Table 2: Satellite subsystem summary for the high capability bus

Subsystem	Capability	Mass	Power	Size
		[kg]	[W]	[cm]
Attitude Determination & Control (ADC)	0.1°-1° attitude control	23.3	49.5	35 x 35 x 22
Telemetry, Tracking, & Command (TTC)	274 Mbs, Ku-band transmitter	10.6	64.4	25 x 25 x 15
Navigation & Guidance (NG)	12 channel GPS receiver	0.0	0.8	7.0 x 4.5 x 1.0
Command & Data Handling (CDH)	Plug-n-play USB architecture	15.2	50	34 x 25 x 20
Power Management (PM)	1500 W, 3J array, PPT system	54.6	253	72 x 23 x 21
Structure	Al Honeycomb Panels	38.6	n/a	52 x 40.5 x 71
Propulsion	Not applicable	0	0	0 x 0 x 0
		142.32	**417.7**	**52 x 40.5 x 71**

There are a few important points to note from the tables above. First, the control scheme for the ADC system is based on a momentum bias system with magnetic torque rods providing additional control for the low capability bus (LCB) and a three axis control system for the high capability bus (HCB). Second, a peak power tracking (PPT) control system is used for power regulation. The advantage of a PPT system is that it acts like a low impedance power supply making design integration a simple task. Finally, because of the short mission life, an onboard propulsion system was not included in the system sizing. It is assumed that the orbit altitude will be high enough to meet mission requirements without additional station keeping.

For the focus of this paper, the capabilities of the bus are somewhat inconsequential. The key parameters here are the power and size for each subsystem and the overall bus. With this information, first level requirements for the TCS can be defined, and overall system architectures can be evaluated. For example, the overall size of the bus yields the radiator space available for heat rejection and, when coupled with the system's surface properties, provides the external heat load that must be managed. Also, the size and power for each of the subsystems provides the base plate area available to transfer heat from the components to the satellite bus and the power density that must be managed at a subsystem level.

Since the focus of ORS is to deploy a spacecraft within six days of call-up, the primary design drivers of the system are modularity and ease of integration. To achieve this goal, the satellite or the subsystem components will have to be on hand for rapid integration and launch; however, their state of pre-integration is still open for debate. There are three primary options. The first is the more flexible option in which the components are on hand so that the satellite can be quickly assembled to meet the needs of the mission. The second and faster option would be to have the satellite preassembled so that it is ready for integration to the launch vehicle. The final option is a combination of the two where the subsystems are preassembled and then integrated into the satellite structure based on mission needs. Because this option provides both flexibility and speed to some degree, it was used as the integration strategy. The actual layouts for the LCB and HCB are shown on the figures below. In addition, the figures show the face that is reserved as the interface plane between the bus and the payload. At this location there is no heat transfer between the bus and the payload or between the bus and the external environment

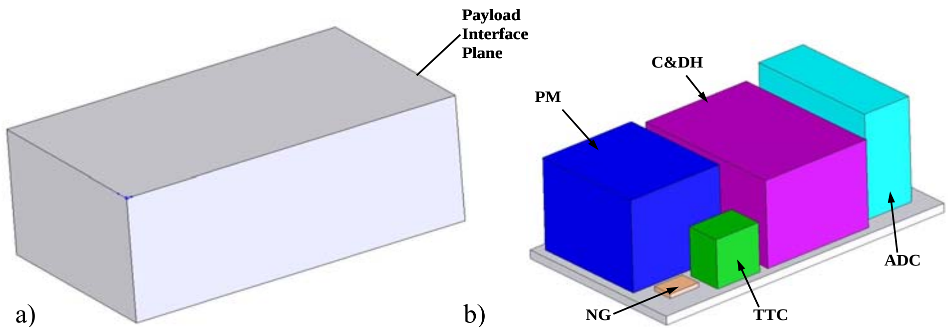

Figure 1: Solid models of the LCB a) exterior and b) subsystem location and orientation

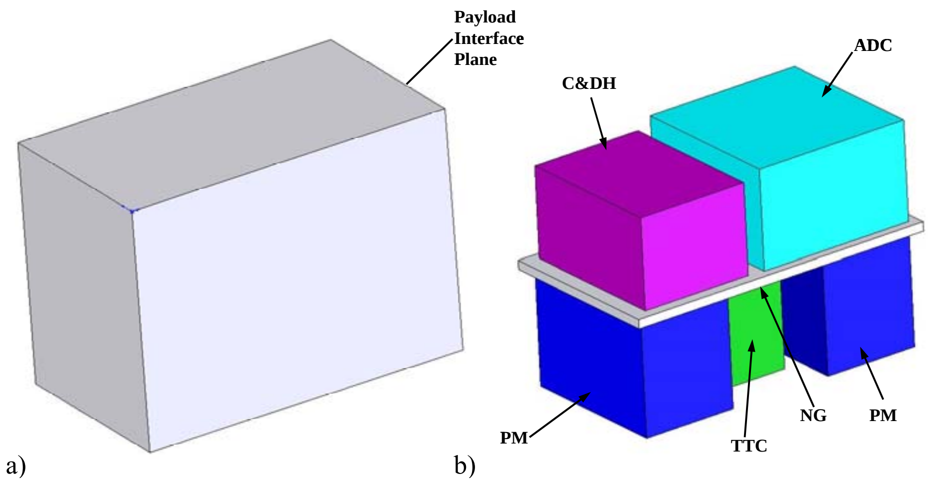

Figure 2: Solid models of the HCB a) exterior and b) subsystem location and orientation

III. External Environment

For most spacecraft, the thermal environment and the external heat loads are determined from the specific orbit for the mission, the orientation of the spacecraft, the surface properties, and the size of the system. From these, the worst hot and cold case conditions are determined. Unfortunately, none of these parameters are clearly defined for ORS missions. The only constraining assumption that can be made is that the orbit regime is limited to low Earth orbits. Since specific orbits are largely unknown for ORS, the TCS must be adaptable to all low earth orbits. For simplicity, only circular orbits were evaluated.

Using these constraints, the absolute worst hot and cold case conditions were determined. The worst hot case condition is shown on Fig. 3a and is defined below[3,4].

- Orbit beta angle is 90°.
- Eclipse duration is zero.
- The panel with the largest surface area is always nadir facing.
- The panel with the second largest surface area always faces the sun.
- Solar flux is 1414 W/m^2.
- Earth IR is 275 W/m^2.
- Albedo coefficient is 0.57.
- The side reserved for the payload faces space. That side does not radiate heat to space.

The worst cold case condition is shown on Fig. 3b and is defined below [3,4].

- Orbit beta angle is 0°.
- Eclipse duration is 43%.
- The panel with the smallest surface area is always anti-nadir.
- Solar flux is 1322 W/m^2.
- The side reserved for the payload is nadir pointing, so there is not an Earth IR or Albedo heat load.

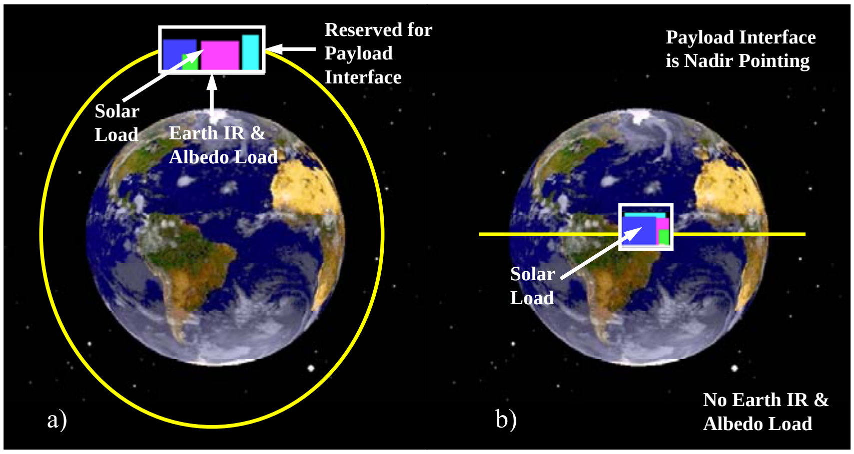

Figure 3: Worst case orbits and orientations a) hot case and b) cold case

It is important to note that these are theoretical worst case conditions. For example, it is unlikely that both the Albedo and Earth IR maximum heat loads will occur at the same time. The Albedo heat load increases with orbit inclination; whereas, the Earth IR heat load increases with decreasing orbit inclination. The theoretical worst case scenarios were chosen to provide confidence in the design and to add a significant amount of margin for most orbits.

It is also important to address the transients of the low Earth orbit (LEO) environment. Because of the low altitudes and short orbital periods, the LEO environment is dynamic and creates special difficulties for the thermal engineer. A LEO spacecraft only sees a small portion of the Earth. As it orbits, it is exposed to rapidly changing environmental conditions as it passes over various geographical features and local time zones, which significantly affect Earth IR and Albedo heat loads. In addition, eclipse times can vary from nearly half of the orbital period to zero. As a result, the thermal capacitance of the system is important, especially for lightweight components. The focus of this effort is on the core bus structure and not external component such as solar arrays or antennas. For this reason, orbit averaged values were used because of the large thermal capacitance associated with the bus.

Finally, before evaluation of the system can be initiated, there is one remaining topic that must be addressed. The fundamental purpose of the TCS is to maintain components within their acceptable operating and survival temperature limits. These limits are wide ranging and component dependent. To ensure that all of the components are within their operating temperature limits the components with the tightest temperature range were used to define the temperature requirements of the system. For this case, they were the momentum wheel and Lithium ion batteries, which have an operating temperature range from 263 to 313 K. Finally, a 10 K margin was added to the temperature limits so that the target design limits are from 273 K to 303 K.

IV. Energy Balance

Essentially, the primary task of the thermal engineer is to balance the thermal energy of the satellite to ensure all of the internal components remain within their acceptable temperature limits during the worst hot and cold cases. Often times a simple energy balance analysis is used to begin the analysis of the system from a very simplistic and general view. The single node energy balance equation provides a basic understanding of the problem and provides a starting point to begin thermal design. For example, it is used to determine whether or not the satellite has enough surface area to maintain its temperature within acceptable limits for the hot case. In addition, it can be used to determine the required survival heater power to maintain the temperature within acceptable limits for the cold case.

The actual temperature of space is 4 K; however, as a first order approximation the temperature of space can be assumed to be 0K. Substituting in expressions for the heat loads, the energy balance equation is[5]:

$$\varepsilon\sigma A_{rad} T_s^4 = \varepsilon A_s F_{s,e} I_{EIR} + \alpha A_{\perp} I_{sun} + \alpha a A_s F_{s,se} I_{sun} + Q_{Internal} \quad (1)$$

where ε is the emissivity of the spacecraft, σ is the Stefan-Boltzmann constant [W/m^2-K^4], A_{rad} is the radiator surface area [m], T_s is the average temperature of the spacecraft [K], A_s is the surface area [m], $F_{s,e}$ is the view factor between the spacecraft and the Earth, I_{EIR} is the intensity of the Earth IR, α is the surface solar absorptivity, $A_{\perp}$ is the area perpendicular to the sun [m], and I_{sun} is the solar heat flux [W/m^2], a is the Earth albedo coefficient, $F_{s,se}$ is the view factor between the spacecraft and the sunlit Earth, and $Q_{Internal}$ is the internal heat generation [W]. This equation provides a first order approximation of the radiator area need for the hot case and the heater power needed for the cold case.

A. Energy Balance for the Low Capability Bus

By rearranging Eq. 1 and solving for A_{rad}, the radiator surface area required to keep the satellite below the maximum operating temperature during the hot case condition can be calculated. The cold case temperature is also determined using Eq. 1 by solving for T_s. If the temperature exceeds the lower temperature limit, survival heaters must be used to provide additional heat. Using Eq. 1 to determine the radiator area and the survival heater power for the satellite provides a first order approximation to size the TCS. It also provides a tool to quickly eliminate thermal control schemes and hardware that will not be applicable to the problem.

For the first order approximation of the energy balance, the internal heat load for the hot case, which was summarized on Table 1, is 147.0 W. As for the cold case, the internal heat load was assumed to be 50 W, which represents the minimum power level required during survival operations. Next, it was assumed that the surface was painted white, and only five surfaces were available for radiation to space. The remaining surface was reserved as the interface surface for the payload. An emissivity of 0.88 and an absorptivity of 0.22 were used for white paint. The inputs into the energy balance equation are summarized below.

Table 3: Summary of the inputs for the energy balance equation for the LCB

	Hot Case	Cold Case
Eclipse Percent	0	0.43
Solar Constant [W/m^2]	1414	1322
Albedo Coefficient	0.57	0.18
Earth IR [W/m^2]	275	218
Internal Heat [W]	147	50
Temperature Limit [K]	303	273
Area $\perp$ to Sun [m^2]	0.192	0.109
Area $\perp$ to Earth [m^2]	0.288	0

Using the energy balance equation and the parameters above, the radiator area required to keep the bus below 303 K was 0.76 m^2 and the resulting cold case temperature was 204.6 K. The total available radiator area of the bus was 1.07 m^2. If the surface area was increased to the total available radiator area, the hot case temperature was reduced to 278.3 K, and the cold case temperature was reduced to 187.9 K. To increase the cold case temperature to acceptable levels, a survival heater power of 240 W would be required. A passive thermal control system incorporating survival heaters would satisfy the thermal requirements. However, an active system might be needed because of the large survival heater power requirement.

B. Energy Balance for the High Capability Bus

For the high capability bus, the internal heat loads for the hot case and cold case were 417.7 W and 50 W, respectively. Again, it was assumed that only five sides of the satellite were available for radiation to space, the surface finish was white paint, and the temperature limits remain unchanged. All of the input values are summarized on Table 4. Following the same process as before, the radiator area required to keep the bus below 303

K was 1.59 m^2; however, the available radiator area was only 1.52 m^2. The result was a hot case temperature of 306.5 K. The cold case temperature was 183.0 K, and the survival heat power was 360 W. Because the system was already above the maximum temperature limit, supplemental radiator area will be required.

Table 4: Summary of the inputs for the energy balance equation for the HCB

	Hot Case	Cold Case
Eclipse Percent	0	0.43
Solar Constant [W/m^2]	1414	1322
Albedo Coefficient	0.57	0.18
Earth IR [W/m^2]	275	218
Internal Heat [W]	417.7	50
Temperature Limit [K]	303	273
Area ⊥ to Sun [m^2]	0.287	0.211
Area ⊥ to Earth [m^2]	0.392	0

V. Thermal Control System Architecture

As noted previously, the subsystems will be housed in separate enclosures to provide both speed and modularity. For subsystems to be interchangeable, electrical, mechanical, thermal, and software standards will have to be developed similar to those in the computer industry. Since thermal design is separated into two parts, overall bus design and component specific design, a natural breakpoint occurs at the interface between the bus and the subsystems. Rather than having to specify interface standards for every type of component, standards would only have to be created for the subsystem enclosure/bus interface. By separating at that location, the subsystem supplier would be responsible for developing the thermal design of the components inside the enclosure; whereas, the system integrator would be responsible for developing the overall thermal control of the bus. The interface between the bus and the subsystems enclosures would be dictated by a thermal design standard that both parties would have to follow. The remainder of this paper focuses on the overall thermal control architecture of the bus and not the specific components within the enclosures.

Two different architectures were evaluated. The first was a traditional semi-passive approach. In this approach, the surface properties, radiators, and interface materials were tailored to ensure proper operation during the hot case. Survival heaters were then sized for the cold case to maintain the proper temperature limits. In the second approach, a thermal switch architecture was used. Here the properties were tailored to ensure maximum heat transfer during the hot case, but minimal heat transfer during the cold. The goal was to size the switch such that the need for survival heaters would be eliminated.

A. Semi-Passive Approach

Completely passive systems are preferred by designers when possible because they are simple, reliable, lightweight, and relatively inexpensive. Unfortunately, purely passive systems are typically not robust and cannot meet the needs of many systems. They definitely will not meet the needs for ORS. More commonly used are semi-passive systems. In this approach, passive technologies are used to transfer heat from the components to the radiators. The radiators are then sized for peak power dissipation, and survival heaters are used to ensure the components do not get too cold during periods of off-peak power dissipation. For systems that have high power densities or experience large fluctuations between hot and cold case conditions, traditionally active systems must be used to maintain adequate cooling and to reduce survival heater power.

The bus structure and the subsystem enclosures were modeled in Thermal Desktop (TD) using aluminum honeycomb panels. Conservatively, a joint conductivity of 110 W/m-K was assumed for bare aluminum interfaces[4]. However, for high power density subsystems, an RTV interface was required to enhance conduction through the enclosure base plate to the satellite structure. For these joints a conductivity of 435 W/m-K was used. As for the boundary conditions, the internal heat loads for each subsystem were evenly distributed over all six surfaces of the enclosure. The external loads were applied using surface heat loads. The solar loads for the hot and cold cases were 312 W/m^2 and 166 W/m^2, respectively. The combined Earth IR and Albedo load for the hot case was 414 W/m^2. White paint with an ε of 0.88 and an α of 0.22 was used for the exterior of the satellite. RadCAD was used to

calculate the radiation exchange factors with space. Radiation within the bus was included in the calculations. The interior surfaces were painted black to enhance radiative heat transfer within the satellite.

Using the constraints outlined above for the internal and external heat load, the LCB and HCB systems were modeled and a simple semi-passive thermal control system was designed. By tailoring the surface properties of the satellite, the interface interstitial materials, and the location and orientation of the components, the TCS was able to maintain the majority of the components below the strict temperature limit of 303 K and all of the components well within their individual temperature limits for the LCB.

The design of the HCB was more complicated in that three deployable radiators had to be added to achieve proper cooling. The radiator locations are shown on Figure 4 and are 0.35 m by 0.405 m. In many designs, Multi-Layer Insulation (MLI) is used to minimize the external heat load and the need for deployable radiators. However, for this scenario, the reduction in the external heat load does not counter act the lost radiator area, and deployable radiators were still required. Also, the use of MLI was not considered practical for ORS operations because of its complicated fabrication process, high touch labor, and fragility. Also, since orbits, missions, orientations, and components are unknown, its pre-application to the structural panels is impractical.

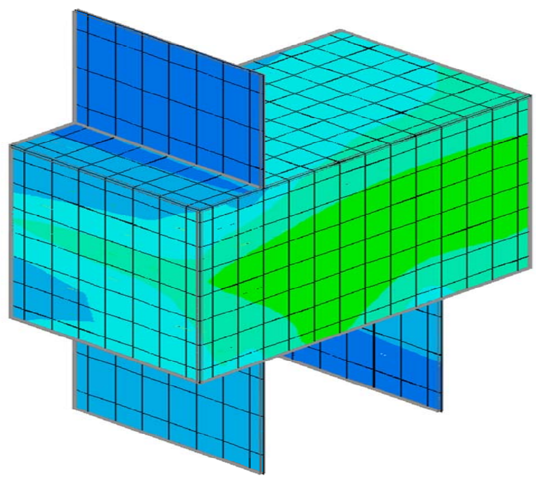

Figure 4: Location of the deployable radiators for the HCB

As for the cold case, the temperatures of all of the components were well below the minimum temperature limit without supplemental survival heater power. To maintain the bus within the baseline temperature limit, an additional heater power of 150 W was needed for the LCB, which was higher than the total power consumption for the hot case. The HCB required 165 W of survival heater power.

The high survival heater power requirements are a result of the drastic change in the external load between the hot and cold cases. The same problem was reported by Barton, where survival heater power was 63% higher than the component operating at full load[6]. As for the worst hot and cold case conditions defined here, it is important to note that it is impossible for both cases to exist for the same orbit. For a more realistic analysis, the worst hot and cold cases were separated by orbit and are outlined below. For each different orbit, the surface properties were tailored and then the heaters were sized.

1a. Worst Case for Hot Orbit: Same as before; results are unchanged
1b. Cold Case for Hot Orbit: Beta angle of 90°, minimum power output, and an orientation with the payload facing the Earth and the smallest adjacent side receiving the solar load
2a. Hot Case for Cold Orbit: Beta angle of 0°, maximum power, and an orientation with largest panels exposed to the solar, Earth IR, and albedo loads
2b. Worst Case for Cold Orbit: Same as before

For case number one for the LCB, the satellite exterior was painted white, and the survival heater power required was reduced to 30 W. For case number two, the exterior was painted green, which increased the solar absorptivity

to 0.57. The emissivity was unchanged. The survival heater power needed to maintain the minimum temperature was reduced to 40 W. As for the HCB, the survival power requirements were reduced to 115 and 90 W, respectively.

B. Thermal Switch Approach

Thermal switches provide control by switching between high and low conductivity regimes at a specific set point. When the temperature is below the set point, the switch is open, and its thermal conductivity is low. When the temperature is above the set point, the switch closes the conduction path. This is typically done by placing the hot side and cold side of the switch in intimate contact. When the temperature of the component drops below the set point, the surfaces are separated, and conduction is minimized. Most heat switches are either on or off. As a result, they can handle rapid fluctuations in power. However, they can also be damped by thermal capacitance and mechanical properties to avoid overshoot and high cycling rates.

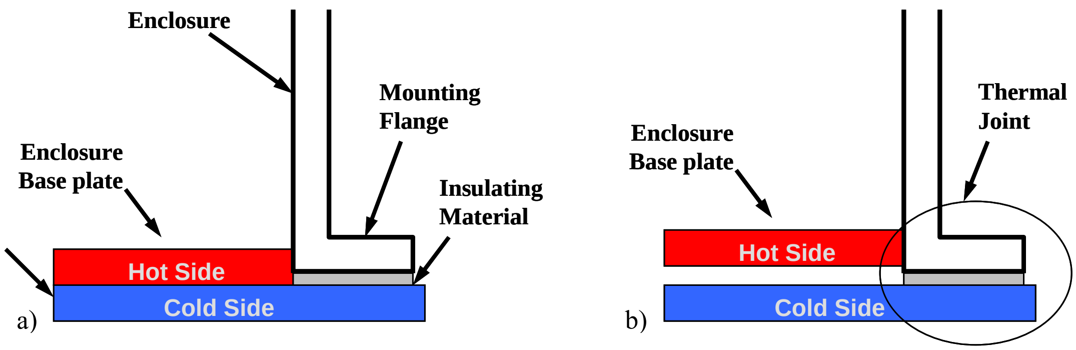

Figure 5: Thermal switch operation a) hot case where the switch is closed and b) cold case where the switch is open

Heat switches can either be mounted between the structure and the radiator or between the structure and the component. When mounted between the structure and the radiator, the heat switch maintains the entire spacecraft at a specific temperature, but the radiator temperature is allowed to fluctuate. As a result, the radiator can be oversized without affecting the overall temperature of the satellite, which reduces the need for heaters. Also, radiators can be designed quickly with robust margins. When mounted between the component and the structure, the structure is allowed to vary with the radiator, but individual component temperatures are controlled. This provides more flexibility than mounting between the structure and radiators because different set points can be used for each heat switch. The advantage of this type of system is that thermal control is only applied to those components that need it. Also, the need for MLI is essentially eliminated because the radiator area is decoupled from the components when the switch is off. The disadvantage is the added system mass. Using heat switches results in a completely different design approach than traditional methodologies. For the system architectures shown in Figures 1 and 2, the optimal locations for the thermal switches are between the subsystem enclosures and the satellite structure.

Although they have important advantages, thermal switches have never been used for whole satellite thermal control, but rather for niche cryogenic sensor applications. One reason is they are limited significantly in size and capability as an inherent property of how they function. To work, they must minimize conduction when open, which means an absolute minimal mechanical support. When closed, they need maximum contact conduction. These two opposing requirements have caused failures that have prohibited their use in general.

For a satellite in LEO, the external heat load is highly cyclical as it passes in and out of the Earth's shadow. Combined with the very short orbit time and component duty cycles, heat switches must endure a large number of switching cycles, which makes fatigue failures problematic. In addition, the contact surfaces must remain clean to maximize conduction. Any contamination significantly reduces performance. Surfaces that have managed to stay clean for more than 10's of cycles have cold welded during thermal vacuum testing. Cold welding occurs when the surfaces become molecularly bonded with a force stronger than what is available to separate them again.

This type of architecture is essentially a variable heat rate architecture and can be achieved by other means than just thermal switches, such as loop heat pipes and variable emissivity radiator systems. The key is to be able to change the heat transfer rate between the hot and cold cases. However, as noted before, there are significant

technical challenges that still face thermal switches. Instead of basing the analysis on a single technology, a more general analysis was conducted to determine the switching requirements for such architectures.

For thermal switches, the critical design parameters are the temperature of the cold side of the switch, the temperature on the hot side of the interface, and the heat transfer rate during the cold case. The hot side and cold side temperatures are somewhat dictated by the characteristics of the system. The temperature on the hot side of the switch is dependent the contact area when the switch is closed, the internal power dissipation, and the cold side temperature. The cold side temperature is dependent on the temperature of the radiator, the heat transfer path, and the power density of the subsystem. For the hot case design scenario, the cold side temperature is the design driver. It must be low enough to ensure proper cooling, but it must not be so low that it causes high cycling rates. For the cold case, the design driver is the heat transfer rate through the switch when it is open. For most switches, this is dependent on the conductivity of the thermal joint.

As shown in the previous section, an acceptable TCS can be designed for the hot case. The problem was the amount of survival heater power required for the worst case cold condition. For this reason, the focus of the design for the thermal switch architecture was on joint conductivity because it is the design driver for the cold case.

Since the temperature on the hot side of the interface is dependent on the system parameters i.e. the contact area, the internal power dissipation, and the cold side temperature, it is difficult to identify a single joint conductivity that would meet the thermal needs for all potential components and subsystems. A very small joint conductivity on the order of 1 W/m^2-K would probably meet the needs of the most systems, but it might not be possible to design a thermal joint with that small of a thermal conductivity. To better gauge the need, the LCB and HCB designs were evaluated. For the LCB, the cold case temperature from the energy balance was 187.9 K. For the HCB, it was 183.0 K. Using the cold case power consumptions and the contact areas for each enclosure, based on the thermal joint above, the joint thermal conductivity required to keep the subsystem temperatures above the lower temperature limit was calculated. The results are presented on Table 5.

Table 5: Joint conductivity required to meet the minimum temperature limit

System	Heat Load [W]	Surface Area [m^2]	Power Density [W/m^2]	K_J [W/m^2-K]
LCB				
ADC	18.5	0.0168	1101.19	12.80
CDH	13.0	0.0236	550.85	6.41
PM	16.2	0.0184	880.43	10.24
TTC	7.4	0.00672	1101.19	12.80
HCB				
ADC	18.5	0.0228	811.40	9.43
CDH	13	0.0236	550.85	6.41
PM	41.2	0.0372	1107.53	12.88
TTC	7.4	0.016	462.50	5.38

To meet the needs of all of the subsystems on the table, a joint conductivity of 5W/m^2-K is required; however, this does not take into account the temperature rise from the enclosure to the component, and a joint conductivity on the order 10W/m^2-K will probably be acceptable. The design or description of such a joint is beyond the scope of this effort.

For architectures based on thermal switches, the performance of the system is based on the conductance ratio of the system. As with all design parameters, there is a trade off. If the conductance ratio is high enough, then survival heaters can virtually be eliminated. However, if the conductance ratio is too high, the system will have a high cycle rate ensuring a fatigue failure. The optimal conductance ratio is system dependent, but ratios on the order of 20:1 to 70:1 are acceptable for a robust operational system.

VI. CONCLUSIONS

This effort was an initial investigation into the system architectures that are best suited for ORS. In that process, preliminary thermal requirements were identified based on the range of satellite capabilities presented. From there,

a traditional semi-passive approach was compared to a thermal switch based architecture. Because of the wide range of components, missions, and orbits envisioned for ORS, a thermal switch based architecture provided a better solution for a one-size-fits-most system because it significantly reduces the survival heater power requirement. However, its success is dependent on developing either passive thermal switches or lightweight, low power active systems suitable for small satellites.

There is one additional disadvantage to this system architecture, which is that the switching component typically adds an additional thermal interface to the system. For radiator panels that are already operating at their limit, adding the additional interface will cause the components to exceed their operating temperatures during the hot case. As a result, radiator area has to be oversized to ensure proper operation, which will add some mass to the system. However, the advantage of a modular, robust system outweighs the disadvantages when a short turn-around-time becomes more important than mass.

The thermal control system poses significant challenges to the goals of ORS. Highly optimized systems will not be feasible on the short time scale dictated for tactical satellites. Instead modular, robust, adaptable systems are required. To meet these challenges, two areas of development are critical. The first is system architecture and design tools. The second is the technologies capable of meeting the requirements dictated by the system architectures.

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